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Canadian Aviation Safety Board

Rapport d'un fait aéronautique civil

Civil Aviation Occurrence Report

Canada

The Canadian Aviation Safety Board investigated this occurrence for the purpose of advancing aviation safety. It is not the object of the Board to determine or apportion any blame or liability.

AVIATION OCCURRENCE REPORT

OKANAGAN HELICOPTERS
BELL 214ST C-GVZO
BIG LAKE, ALBERTA
23 OCTOBER 1985

REPORT NUMBER 85-H50007

SYNOPSIS

The helicopter departed Edmonton Municipal Airport on a visual flight rules flight. The purpose of the flight was to test and document aircraft systems response following simulated inverter failures. Forty minutes after departure, an in-flight breakup occurred. The main fuselage crashed in a wooded area and burned; both crew members were fatally injured.

The Canadian Aviation Safety Board determined that mast bumping occurred, most likely as a result of control inputs during manoeuvring flight, causing the weakened collective sleeve to fail. Although the severity of the mast bumping in itself may have led to the severing of the tail boom, the failure of the collective sleeve may also have contributed to the rotor blade to tail boom contact. The collective sleeve was cracked and weakened by an improper deviation from the authorized manufacturing process.

Ce rapport est également disponible en français.

19 January 1989

TABLE OF CONTENTS

		Page
1.0	FACTUAL INFORMATION	1
1.1	History of the Flight	1
1.1.2	Background	1
1.2	Injuries to Persons	2
1.3	Damage to Aircraft	2
1.4	Other Damage	2
1.5	Personnel Information	3
1.5.1	Flight Crew	3
1.6	Aircraft Information	4
1.6.1	Automatic Flight Control System	5
1.6.2	AC Electrical System Failures	6
1.6.3	Aircraft Certification	7
1.7	Meteorological Information	7
1.8	Aids to Navigation	8
1.9	Communications	8
1.10	Aerodrome Information	8
1.11	Flight Recorders	8
1.12	Wreckage and Impact Information	8
1.12.1	The Site	8
1.12.2	Trajectory Analysis	9
1.13	Medical Information	9
1.14	Fire	9
1.15	Survival Aspects	9
1.16	Tests and Research	10
1.16.1	Engines	10
1.16.2	Main Rotor Blades and Tail Boom	10
1.16.3	Collective Sleeve	12
1.16.3.1	Collective Sleeve Failure	12
1.16.3.2	Failure History	15
1.16.4	Collective Friction Collet	15
1.16.5	Mast Scrape Marks Analysis	16
1.16.6	Main Rotor Mast	16
1.16.7	Transmission Mount Analysis	17
1.16.8	Computer-Generated Flight-Simulation Model	17
1.16.8.1	Introduction	17
1.16.8.2	Main Rotor Clearance Over the Tail Boom	18
1.16.8.3	Results of the Computer Simulation	18
1.16.9	Warning and Caution Panel Lights and Switch Evaluation	20
1.16.10	Main Rotor Blade Examination	21
1.17	Additional Information	23
1.17.1	Mast Bumping	23
1.17.2	Aerodynamic Analysis of Collective Sleeve Failure	24
1.17.3	Test Flights Conducted by the Operator	25
1.17.4	Further Test Flights	26
1.17.5	Operational Control Aspects of the Flight	26
1.17.6	Characteristics of C-GVZO	27

1.17.7	Witnesses	27
1.17.8	Technical Report on the Study of Selected Aspects of the Bell 214ST Certification	27
2.0	**ANALYSIS**	**29**
2.1	Introduction	29
2.2	Blade Strike Analysis	29
2.3	Collective Sleeve Failure	32
2.4	Control Inputs	34
2.4.1	C-81 Computer Simulation	34
2.4.2	In-flight Inputs	35
2.5	Characteristics of C-GVZO	36
2.6	Collective Friction Collet	36
2.7	Main Rotor Blade Delamination	37
2.8	Operational Considerations	38
2.9	Certification Aspects of the Bell 214ST	39
2.10	Flight Recorders	40
2.11	Summary	40
3.0	**CONCLUSIONS**	**42**
3.1	Findings	42
3.2	Causes	44
4.0	**SAFETY ACTION**	**45**
4.1	Action Taken	45
4.1.1	Collective Sleeve Assembly	45
4.2	Action Required	46
4.2.1	Cockpit Voice Recorder/Flight Data Recorder Equipment	46
4.2.2	Conduct of Tests Flights	47
5.0	**APPENDICES**	
	APPENDIX A - BLADE STRIKE LOCATION	A-1
	APPENDIX B - MAIN ROTOR MAST ASSEMBLY	B-1
	APPENDIX C - LABORATORY REPORTS	C-1
	APPENDIX D - GLOSSARY	D-1

1.0 FACTUAL INFORMATION

1.1 History of the Flight

On 23 October 1985, C-GVZO, a Bell 214ST helicopter belonging to Okanagan Helicopters Limited, took off from the Edmonton Municipal Airport on a visual flight rules (VFR)* flight to a designated practice area 15 nautical miles northwest of the airport. On board were two flight crew members who were to carry out a series of test flights involving simulated electrical malfunctions and to document aircraft system responses and aircraft altitude and attitude changes.

Approximately 40 minutes after take-off and at an altitude of about 1,000 feet above ground level (agl), an in-flight breakup of the aircraft occurred. The tail boom was severed, and the main rotor and transmission assembly separated from the aircraft. The main fuselage then rolled into an inverted attitude, struck the ground in a wooded area, and was consumed in an intense post-impact fire. Both crew members were killed in the crash.

The main wreckage of the helicopter was located at lat 53°34'30"N, long 113°44'00"W**, at an elevation of 2,240 feet above sea level (asl). The accident occurred at approximately 1440 mountain daylight time (MDT)*** during the hours of daylight.

1.1.2 Background

On 13 March 1985, a Bell 214ST helicopter belonging to a subsidiary company, Universal Helicopters Newfoundland Limited, took off from an offshore oil rig anchored in Placentia Sound, Argentia, Newfoundland. It was on a night instrument flight rules (IFR) flight to St.John's and was last seen in a climbing left turn, entering an overcast cloud layer at approximately 350 feet asl. After take-off, the crew did not establish radio contact. The helicopter struck the water in an extreme nose-down, left-bank attitude. The cause of this earlier accident, which claimed six lives, is the subject of CASB accident report 85-H50002 and remains undetermined.

* See Glossary for all abbreviations and acronyms.
** Units are consistent with official manuals, documents, reports, and instructions used by or issued to the crew.
*** All times are MDT (Coordinated Universal Time (UTC) minus six hours) unless otherwise stated.

Okanagan Helicopters considered that an interruption of electrical power to the flight instruments may have been a factor in the apparent loss of attitude control evident in the first accident.

In August 1985, Okanagan pilots decided to explore this theory further, and a number of flights using the Bell 214ST C-GVZO were conducted at Churchill, Manitoba. These flights involved a series of left and right climbing turns with either or both inverters switched off. In all cases, the aircraft was crewed by two pilots. The pilot seated on the right flew the aircraft in accordance with IFR procedures in simulated and actual instrument meteorological weather conditions (IMC), using sole reference to the aircraft flight instruments. The other pilot was responsible for selecting one or both inverters off, monitoring the performance of the pilot flying, and ensuring that the aircraft was flown within the flight envelope. The results of these test flights were compiled and submitted to the operator's head office in Vancouver, British Columbia. Subsequently, the company decided to conduct further tests in Edmonton, using C-GVZO.

1.2 Injuries to Persons

	Crew	Passengers	Others	Total
Fatal	2	-	-	2
Serious	-	-	-	-
Minor/None	-	-	-	-
Total	2	-	-	2

1.3 Damage to Aircraft

The aircraft was destroyed by the in-flight breakup, the impact forces, and the intense post-impact fire.

1.4 Other Damage

There was no other damage.

1.5 Personnel Information

	Pilot-In-Command	Co-Pilot
Age	52	57
Pilot Licence	Commercial Helicopter	Commercial Helicopter
Medical Expiry Date	January 1986	April 1986
Total Flying Time	16,396 hr	21,926 hr
Total on Type	171 hr	115 hr
Total Last 90 Days	10 hr	115 hr
Total on Type Last 90 Days	10 hr	27 hr
Hours on Duty Prior to Occurrence	2 hr	2 hr
Hours off Duty Prior to Work Period	12 hr	12 hr

1.5.1 Flight Crew

Both crew members held valid commercial helicopter pilot licences endorsed with Class I, Group IV instrument ratings. They were both fully qualified to operate as company pilots; the pilot-in-command had recently retired from the position of Okanagan chief pilot and was employed as a contract pilot, enabling him to continue the proposed test-flight schedule.

For the purpose of this flight, the pilot-in-command occupied the left seat and is believed to have been documenting the aircraft and instrument response in accordance with the test-flight schedule. Included in the type endorsements on his licence were the Bell 214ST, Bell 212, Bell 206, Bell 205, Sikorsky S76, Sikorsky S58T, and Sikorsky S61 type helicopters.

The co-pilot occupied the right seat and is believed to have been the pilot flying the aircraft. His licence type endorsements included the Bell 214B, Bell 214ST, Bell 212, Bell 206, Bell 205, Sikorsky S76, Sikorsky S61, and Aerospatiale AS332 type helicopters.

Both crew members were senior pilots, each holding the qualification of Transport Canada approved company check pilots, with extensive offshore and instrument flying experience.

1.6 Aircraft Information

Manufacturer	Bell Helicopter Textron
Type	Bell 214ST
Year of Manufacture	1982
Serial Number	28112
Certificate of Airworthiness	Valid
Total Airframe Time	2,039 hr
Engine Type (2)	General Electric CT7-2A
Rotor Type	Bell 214-015-300-105
Maximum Allowable Take-off Weight	17,500 lb
Recommended Fuel Types	Jet A, Jet A-1, Jet B

The helicopter was certificated in 1982 under Federal Aviation Administration (FAA) type certification as a transport-category rotorcraft in accordance with Federal Aviation Regulation (FAR) 29 and the IFR standards dated 15 December 1978. A certification validation program was carried out by Transport Canada in 1982 to permit issue of a Canadian Type Approval for this helicopter type.

The helicopter was purchased by Okanagan Helicopters Limited in 1983, imported into Canada, and was issued a Certificate of Airworthiness. The aircraft was primarily used in offshore oil and gas exploration.

The helicopter was maintained in accordance with existing regulations and Transport Canada approved procedures. All modifications, mandatory Airworthiness Directives (AD) and Maintenance had been complied with.

Prior to the accident, this helicopter had a history of collective collet failures, and, at 1,659 airframe hours, the main rotor mast had been replaced, primarily because the mast friction sleeve was worn beyond limits. The in-flight rotor tracking system on the aircraft had been previously disconnected because of accelerated wear on the upper bearing drive link, and was finally removed in December 1984. There was also a fatigue crack in the right-hand clevis pylon support of the nodal beam mounting system, which was being monitored by Okanagan Helicopters, under advice of the manufacturer. Prior to the last take-off, a slight vertical vibration at the rate of one vibration per main rotor revolution had been reported.

1.6.1 Automatic Flight Control System

The Bell 214ST uses a complex automatic flight control system (AFCS) which includes a stability control augmentation system (SCAS), an attitude altitude retention system (AARS), and a fly-by-wire elevator system (FBW). The SCAS reduces pilot workload by improving stability and handling qualities. The AARS provides the pilot with hands-off cruise flight capability. The FBW elevator system is designed to enhance the flight handling qualities in relation to the stability of the aircraft in the pitch axis.

The FBW system improves handling qualities of the helicopter by adjusting the elevator angle in flight to provide static speed stability, to reduce attitude trim changes with collective movement, and to minimize trim changes with centre of gravity travel. On-board electronic sensors measure airspeed, pitch cyclic position, and collective lever position. These signals are then sent to the FBW amplifier to position the elevator accordingly. The FBW system incorporates two independent self-monitored systems, each driving an independent, electrically operated servo, which in turn causes the angle of attack of the elevator to change according to the in-flight requirements. In the event of a complete power interruption or system shutdown, the elevator will freeze at the position immediately prior to the failure. This FBW system is unique to the Bell 214ST; other civilian helicopters which employ movable horizontal elevators have the elevator controls linked mechanically rather than electrically.

The SCAS is a three-axis (pitch, roll, and yaw), limited authority, stabilization system using a series of electro-mechanical actuators. Each axis of the system uses signals from rate gyros and control motion transducers (CMT) to provide drive signals to the actuators. In addition, a turn coordination feature is provided through the yaw SCAS actuator based on the roll attitude signal from the right-hand-side pilot's attitude gyro.

The AARS is a two-axis (pitch and roll) system using signals from both the pilot and co-pilot attitude gyros, the altitude controller, the airspeed and collective position signals from the FBW computer, and heading error from the horizontal situation indicator (HSI). The three functions provided by the AARS are attitude hold, heading hold, and altitude hold; the latter two may be selected simultaneously by the pilot.

With the AARS engaged, the attitude of the helicopter may be trimmed by either the beep-trim function or the force-trim release function. The controls for both functions are located on the pilot and co-pilot cyclic controls. The beep-trim function is activated from a five-position switch on the top of the cyclic control. Beep-trim input will produce an aircraft attitude change which corresponds to the direction in which the beep-trim switch is moved. The pilot may also depress the force-trim release switch on the left side of the cyclic control and manually make a cyclic control input to achieve an attitude change.

If the cyclic control is moved without the use of the beep-trim switch, one or both of the AARS axes will temporarily disengage. When the cyclic control is returned to the original position, the channels which were temporarily disengaged are re-engaged, and the aircraft will return to its original trim attitude at a rate of two degrees per second.

1.6.2 AC Electrical System Failures

In the event of either a number one or number two inverter failure, AC power will normally be provided automatically from the remaining functional inverter within 1.3 seconds after the inverter failure.

A failure of the number two AC power inverter will cause the SCAS to disengage, with the SCAS actuators all returning to their centre positions; however, upon AC power failure, the short-term stability functions in the pitch and roll axes of the SCAS will be provided by the AARS, if engaged. In addition, the SCAS is designed to re-engage automatically if the remaining functional inverter assumes the load responsibility of the failed inverter within 1.3 seconds and if the SCAS self-diagnostic tests are valid. However, if the remaining inverter takes longer than 1.3 seconds or the SCAS self-diagnostic tests fail, the pilot must manually re-engage the SCAS.

The rotorcraft flight manual (RFM) cautions that with the SCAS inoperative, the use of the force-trim release button will result in the discontinuation of AARS functions for as long as the button is depressed. The RFM recommends that the beep-trim switch be used to make pitch and roll attitude changes and that care should be exercised when using this switch to prevent inadvertent pilot stick motion or overcontrolling.

In the event of a dual-inverter failure, neither the SCAS nor the AARS will function. Without an AFCS function, the aircraft is less responsive to pilot

control inputs, and the risk of overcontrolling is increased. In addition to the reduced stability of the helicopter with the SCAS inoperative, as the electrical power to the SCAS is interrupted and if the SCAS actuators are displaced from their centre positions, the actuators re-centering will result in a flight control input to the main rotor and could cause the attitude of the aircraft to change. For example, depending on which side of centre the actuator was at the time, if the SCAS power supply was interrupted with the helicopter in a left turn, the effect of the re-centering roll actuator could be to increase the left roll attitude.

1.6.3 Aircraft Certification

The Bell 214ST is certificated in Canada for two-pilot, day/night VFR and IFR, Transport Category B operations in non-icing conditions. For day or night VFR flight, operable SCAS and FBW are required. For IFR operations operable SCAS, AARS and FBW are required.

1.7 Meteorological Information

A ridge of high pressure over southern British Columbia was influencing the weather in central and southern Alberta at the time of the occurrence. The upper air mass was dry and unstable with towering cumulus tops developing to about 20,000 feet. The winds from the surface to 2,000 feet agl were about 290 degrees true at 18 to 20 knots.

The actual weather sequence at Edmonton Municipal Airport was 4,000 feet scattered, 25,000 feet thin scattered, 15 miles visibility, temperature nine degrees Celsius, dew point seven degrees Celsius, wind 280 degrees true at 18 gusting to 23 knots, altimeter setting 29.35 inches of mercury. The sky coverage was cumulus one-tenth, alto-cumulus one-tenth, cirrus one-tenth.

There was no indication or report of turbulence in the local area. The weather conditions at the time of the accident were not conducive to the formation of any turbulence greater than light turbulence associated with the tree line and the lake near the accident site.

1.8 Aids to Navigation

For the purpose of this flight, no radio aids to navigation were required.

1.9 Communications

There were no known radio transmissions between the helicopter and any communications facility after departure from the Edmonton Municipal Control Area.

1.10 Aerodrome Information

Not applicable.

1.11 Flight Recorders

The helicopter was not equipped with a flight data recorder (FDR) or a cockpit voice recorder (CVR), nor was either required by regulation.

1.12 Wreckage and Impact Information

1.12.1 The Site

The fuselage struck the ground in an inverted position. The impact forces and the intense post-crash fire destroyed most of the cockpit and cabin.

The wreckage trail extended over one and a half miles in a northwest-southeast direction. Pieces of the tail boom and heavier main rotor blade material were in the northwesterly area with the lighter blade pieces and other material scattered to the southeast. The main rotor and transmission assembly struck the ground about 260 feet southwest of the fuselage and the tail rotor pylon was located 230 feet to the southeast.

Some components including the drive shaft between the transmission and the combining gearbox and the two FBW elevator actuators were not located. Without these actuators, the final position of the FBW elevator could not be determined.

1.12.2 Trajectory Analysis

Based on the wreckage distribution and wind speed of 18 to 20 knots derived from meteorological data, a computer-aided trajectory analysis was performed, and the following conclusions were made:

1. The aircraft broke up at an altitude of about 925 feet agl.

2. The wind direction was constant at 290 degrees true.

3. The wind caused light pieces to drift up to 1,800 metres.

4. The aircraft speed was approximately 80 knots.

1.13 Medical Information

Both crew members had recently undergone aviation medical examinations and were assessed fit to the Department of Transport Category 1 standards. Post-mortem examination and toxicological testing did not reveal any evidence of incapacitation or physiological problems prior to the accident.

1.14 Fire

There was no evidence of pre-impact fire. Following ground impact, the fuselage was substantially damaged by the post-crash fire.

Members of the county volunteer fire department responded to the alarm, but, by the time they arrived, the fire was burning intensely. They controlled the spread of the blaze and assisted with the recovery of the bodies.

1.15 Survival Aspects

This was not a survivable accident. The in-flight breakup, particularly the separation of the main transmission and rotor assembly, precluded any possibility of crew survival.

1.16 Tests and Research

1.16.1 Engines

The two General Electric CT7-2A engines were dismantled with the assistance of a representative from the engine manufacturer. It was determined that the engines were serviceable and operating normally at the time of the accident.

1.16.2 Main Rotor Blades and Tail Boom

The main rotor system is two-bladed, and the blades are designated 'red' and 'white'.

The blade strike sequence was of an extremely short duration, approximately two seconds or 20 blade passages, and of these, only seven strikes have been identified.

Examination of the tail boom pieces indicated that multiple main rotor blade strikes occurred while the aircraft was airborne. The determinable blade strikes on the tail boom have been referenced from 'A', at the most aft location, to 'G', at the most forward tail boom location (see Appendix A). However, actual blade strike sequence is not to be implied by this alphabetical identification.

The severe damage to the tail boom made it difficult to measure accurately some blade strike angles. However, careful reconstruction of the tail boom revealed that there were two separate blade strike marks, identified as 'A' and 'B', to the top of the tail rotor drive shaft cover. The cover was crushed and folded and showed signs of having been torn away violently. Both blade tip caps showed heavy impact damage, with several paint transfer marks; however, the aft edge of each blade tip was unmarked. There was also a small area of circumferential scoring on the tail rotor drive shaft coincident with part of the crushed cover. A blade angle analysis was carried out and indicated that the blade tip angle on contact with the drive shaft cover was approximately -12 degrees.

The blade strike 'C', which was near the 42-degree gearbox, severed the tail rotor drive shaft. This strike passed through the tail boom and cut down to seven inches below the tail rotor drive shaft deck. The blade had entered low on the left side of the tail boom and exited approximately on the centre line of the top panel of the tail boom. It could not be determined which blade made this strike.

The 'D' strike was a very heavy hit deep into the tail boom. The angle of this strike indicated that the rotor disc had teetered aft and to the left about 20 degrees. This strike was assigned to the red blade because of the paint transfer evidence and blade and tail boom damage matching.

There was a deep contact mark on the leading edge of the white blade which matched with the damaged tail rotor drive shaft coupling at tail boom station 243.7.

The scratch marks observed on the white blade were equally distributed on the top and bottom surfaces.

At location 'E', the blade entered the tail boom after diagonally cutting through the left side FBW elevator. The blade exited the tail boom at a point near the trailing edge of the right-side FBW elevator and did not pass through any significantly massive components. Strike 'E' is assigned to the white blade because of the deep contact mark on the white blade leading edge mating with the damage at tail boom station 243.7.

There was a severe impact damage site on the leading edge of the red blade that matched with the damaged FBW elevator spar.

The blade strike at 'F' caused severe impact damage to the tail boom and FBW elevator spar. The damage to the spar and both elevators indicated a massive impact sufficient to drive the spar through the fuselage and into the right elevator, causing it to buckle. The orientation of this blade strike indicated that the transmission was separated from the aircraft. The tail boom contact angles indicated that the rotor hub was out of position, displaced up and tilted aft and to the left of the tail boom. The severe damage site on the red blade that coincides with the damage on the FBW elevator spar and both elevators enables the identification of the red blade as being responsible for this blade strike.

The tail boom strike forward of strike 'F' was at an extremely high blade pitch angle. This was evidenced by matching blade and main fuselage contact marks which indicated almost a flat blade slap rather than a cutting action. There were areas on the red blade of paint transfer from this section of the tail boom, which enabled identification of strike 'G' to the red blade.

Each blade had a built-in twist of -8.32 degrees; that is, a blade angle of +8.32 degrees measured at the root would be zero degrees at the tip. In consideration of the aircraft's weight, centre of gravity, assumed airspeed, and flight profile of a climbing, left turn at 80 knots, prior to breakup, it has been estimated that the collective control position for C-GVZO would have been about 46 per cent collective position which would give a root pitch angle of +11.75 degrees or +3.43 degrees at the tip. Reducing the collective pitch setting to 6 per cent would give a root angle of +4.85 degrees or -3.47 degrees at the tip. Similarly, it has been calculated that the stabilized blade root angle after collective sleeve failure would be +8.35 degrees or -0.03 degrees at the tip. There is variation of this collective datum blade pitch angle caused by those cyclical blade angle changes determined by the existing amount of cyclic control input.

In order to make contact with the top of the tail rotor drive shaft cover, the mast would initially make contact with the flapping 'soft' stop on the blade yoke and, subsequently, after soft stop failure, make contact with the yoke itself. This progression is witnessed by the contact marks on the mast. The mast indentations caused by the red blade yoke are deeper, indicating that the red blade hit the mast harder.

1.16.3 Collective Sleeve

The collective sleeve (see Appendix B) forms an integral part of the flight control system and serves to transmit the pilot's collective pitch demands to the scissors assembly, which then varies the main rotor blades pitch angle.

1.16.3.1 Collective Sleeve Failure

Initial on-site examination of the wreckage disclosed a failure of the collective sleeve at the lower end. The total time on the part since new was reported to be 2,039 hours.

The CASB Engineering Laboratory was tasked to determine the mode of failure. It was found that the collective sleeve separated in overload and showed the characteristics of failure by tensile overload.

The sleeve was found to have failed at the lower end at the fillet radius to a chromium-plated journal. It was discovered that a large number of defects characteristic of hydrogen embrittlement cracking (HEC) were present in the sleeve. These HEC defects,

although concentrated in some areas, were generally distributed around the full circumference of the sleeve and were found to have been introduced during manufacture. The fracture was coincident with a region of multiple small flaws resulting from the HEC. Approximately 28 per cent of the fracture cross-section was pre-cracked in this manner.

HEC with a number of fish-eye defects, similar to the above-mentioned defects, had been found in one other failed collective sleeve from a Bell 214ST involved in an occurrence in the Sultanate of Oman in February 1984. The collective sleeve flight time was reported to be 123.4 hours, and the component was traced to a particular batch of sleeves. It was determined that the fish-eye defects were caused by an improper deviation from the authorized manufacturing process. During the processing sequence of this batch of sleeves, the manufacturer discovered that the first chromium plating operation was unacceptable. In order to remove the inferior plating, the sleeves were partially immersed in an inhibited hydrochloric acid bath. The affected sleeves were reportedly left overnight in the acid before being replated, rebaked, and sent to Bell. It is speculated that the acid inhibitor was either deficient or depleted and allowed excessive amounts of hydrogen to be generated, which then entered the metal, causing hydrogen embrittlement cracking.

An Airworthiness Directive (AD) T84-05-51 was issued on 27 February 1984, requiring a fluorescent magnetic particle inspection (FMPI) of all collective sleeves P/N 214-010-411-001/003 that had a serial number prefix of "A19." Inspection requirements, based on the life of the component, were as follows:

0-500	Inspect prior to further flight,
500-1,000	Inspect next 100 hours, and
1,000 over	Inspect at regularly scheduled overhaul intervals of 2,500 hours.

The sleeve on C-GVZO, to which the above AD applied, had accumulated 1,197.4 hours at the time the AD was issued. Okanagan Helicopters elected to have the sleeve inspected immediately rather than wait for the next scheduled overhaul. The aircraft logbook showed that this inspection was performed on 28 February 1984 by a company contracted by Okanagan.

The FMPI examination produced no indications of cracking; however, it has subsequently been established that the inspection was conducted prior to the receipt of the Bell Alert Service Bulletin (ASB) 214ST-84-17,

issued 12 March 1984. The performed inspection was deficient with respect to the preparation of the surface of the sleeve; specifically, the paint stripping required by the ASB for the chromium to cadmium-plated radius had not been carried out. Bell had identified this radius to be the region most susceptible to HEC. Tests were subsequently conducted by the CASB Engineering Laboratory on the outside surface of the failed sleeve, adjacent to the fracture, in order to verify Okanagan's previous inspection findings and to validate the prescribed ASB procedures of determining embrittlement. The conclusion drawn from the tests was that the sleeve was cracked at the time of the previous inspection and that the detection of these cracks was prevented by the presence of paint and primer on the surface. Okanagan Helicopters was still in compliance with the AD since the FMPI was not required to be carried out until the component had accumulated 2,500 hours.

Subsequent to this accident, Bell issued ASB 214ST-86-33, dated 20 February 1986, requiring the replacement of all collective sleeves.

A fracture mechanics study was conducted in order to determine whether the HEC constituted a critical flaw, allowing the sleeve to fail under normally applied flight loads. The fracture mechanics calculations showed that 85 per cent of the load bearing area would need to be cracked before critical conditions for failure would occur. This determination was in close agreement with the Oman sleeve failure which was reported to have been pre-cracked by fatigue extension of the original HEC defects to approximately 85 per cent before failure.

The sleeve on C-GVZO had been in service for 2,039 hours with no further growth of the defects by an HEC mechanism. Extension of some of the cracks by fatigue, representing a 1 to 2 per cent increase in the overall size of the flaw, was not considered sufficient to cause collective sleeve failure under limit in-flight load**.

Bell's pre-certification test-flight program calculated the stresses experienced by the sleeve under limit in-flight load. Test data were gathered regarding its ability to withstand loads resulting from manoeuvres within the normal flight envelope.

** Limit in-flight load is defined for aeronautical use as the maximum in-flight load to be expected in service.

1.16.3.2 Failure History

The previous collective sleeve failure on the Oman helicopter, in February 1984, occurred during straight and level flight at a height of 500 feet agl and a cruising speed of 120 knots. This failure showed that the aerodynamic loading on the blades causes them to achieve a relatively low blade pitch angle of about plus eight degrees measured at the root (see section 1.17.2). The helicopter entered a shallow rate of descent and carried out a roll-on landing.

In May 1986, a U.K. registered Bell 214ST, during cruise flight, had a failure of the collective pitch control assembly which has an effect similar to a collective sleeve failure. The crew reported that the transmission torque decreased, resulting in a slow rate of descent. The helicopter attitude control was not affected, and the pilot manoeuvred the aircraft into wind and carried out a successful water landing.

1.16.4 Collective Friction Collet

Examination of the wreckage disclosed a failure of the collective friction collet (see Appendix B) in the scissors and sleeve assembly of the main rotor controls. The collective friction collet is a device that dampens vibrations originating in the main rotor head. These vibrations are transferred to the primary servos and, to a lesser degree, the collective sleeve.

The CASB Engineering Laboratory determined that the failure of the collet occurred at a single location in the region of a slot and at two locations within the mounting flange. Failure at these locations occurred from high-cycle fatigue, which together with the occurrence of fretting damage on the mounting flange, would indicate the presence of cyclic stresses, probably of a vibratory nature from the main rotor head. No material or manufacturing deficiencies were identified which could have contributed to the failure. C-GVZO had a history of friction collet replacement, at least six times in 2,039 hours, with the last change 185.6 hours before the accident. During a six-month period from November 1983 to May 1984, the collet set was replaced three times because of broken segments; the airframe time span for these three replacements was 486 hours.

It was concluded that, although the collet failed at some unknown time before the accident, the pieces were held in place.

The mast friction wear sleeve is a sacrificial metal sleeve bonded to the mast to prevent wear to the mast itself caused by the collective friction collet. The mast change in December 1984, at an airframe time of 1,658.9 hours, was required because the wear on the friction sleeve was beyond limits. At this time, however, it was not necessary to replace the friction collet set. Post-accident investigation revealed that there were several regular grooves in the friction sleeve created by the edges of the individual fingers of the friction collet. There were also additional rub marks seen on the mast friction sleeve.

1.16.5 Mast Scrape Marks Analysis

A number of scrape marks resulting from contact of the collective sleeve with the surface of the main rotor mast were identified. The CASB Engineering Laboratory examined the location of these scrape marks in an effort to relate the position of the marks to any collective pitch control inputs. The wear on the mast friction wear sleeve was also examined.

The mast outer diameter surface exhibited patterns of score marks which matched with the collective sleeve and were attributed to two locations on the inner diameter of the sleeve and the edges of the fracture. The scrape patterns on the mast were consistent with the elastic mast-bending during at least 10 revolutions.

Examination showed that the collective control position at the first contact was between 50 to 60 per cent collective travel and that all subsequent collective positions were in a lower collective position. The upper series of these marks was in such a location on the mast that the collective sleeve fracture surface had to have been at that position which equated to about 40 per cent collective control. The most extreme mark showed that the collective sleeve achieved a position beyond the normal full-down position.

Further scrape marks were seen to coincide with the blade yoke contact position on both sides of the mast and with the pitch horn of the red blade.

1.16.6 Main Rotor Mast

Examination of the main rotor mast showed that the titanium yoke had made hard contact with the main rotor mast and that the mast and yoke were bent. The CASB Engineering Laboratory determined that yoke-to-mast contact (mast bumping - see Section 1.17.1) produced elastic mast-bending, and there were indications that

the transmission and rotor had separated from the aircraft as a combined result of the forces generated by tail boom impact and mast bumping. The forces which caused the flapping stops to fail were part of the forces that caused elastic mast-bending.

1.16.7 Transmission Mount Analysis

The transmission mount nodal beams and support structure were examined to determine the mode and sequence of failure. There was no evidence of fatigue fracture of the nodal mount other than a monitored crack in the right side aft mount to the fuselage structure which did not progress despite the helicopter breakup.

The initial failure appeared to have resulted from damage to the right nodal mount caused by an extreme downward force on the right transmission trunnion. The probable source of the extreme downward force applied to the right transmission trunnion to its nodal mount was tail boom impact combined with yoke-to-mast contact.

1.16.8 Computer-Generated Flight-Simulation Model

1.16.8.1 Introduction

Bell was asked to consider under what conditions the main rotor of a Bell 214ST could contact the tail boom. The parameters used for the flight evaluations conducted by Okanagan Helicopters were provided to the manufacturer so that a computer simulation would approximate the flight believed to have been undertaken at Edmonton. For the purpose of the computer simulation, a Bell 214ST manoeuvre on the Rotorcraft Flight-Simulation Program (C81) was used. The manoeuvre was a 15-degree left bank climbing turn at 80 knots, with a 500-foot-per-minute rate of climb. This particular manoeuvre was chosen as it was used in the Churchill test flights and, according to the Okanagan consultant's test-flight schedule, was one of the manoeuvres yet to be performed and documented in the Edmonton test flights (see Section 1.17.3).

The manufacturer was asked to simulate a dual-inverter power failure for the computer flight analysis. Following the power interruption and as the aircraft pitch-up attitude exceeded 15 degrees nose-up, corrective action was to take place by rapidly lowering the collective. Similarly, as the aircraft responded to the lowered collective and pitched down through 15 degrees nose-down, recovery input was to be initiated with the cyclic control.

It has been determined that, following a collective sleeve failure, the blades would achieve a stabilized root angle of +8.35 degrees. This angle can be related to a specific collective sleeve position corresponding to 26 per cent of collective lever travel.

The computer analysis showed a rapid collective reduction from a cruise position of 46 per cent collective position to the 26 per cent position. This collective reduction occurred within approximately one-half second and produced flapping consistent with either a rapid downward collective movement or a collective sleeve failure.

1.16.8.2 Main Rotor Clearance Over the Tail Boom

The transmission pylon deflection, main rotor aft flapping angle, blade coning, blade feathering, and the geometric relationship between the hub location and the tail boom must be known for the given flight condition in order to determine the main rotor blade clearance over the tail boom. For the Bell 214ST, blade to tail boom contact can occur at zero degrees of blade pitch when the following conditions occur: the main rotor aft flapping angle is 15 degrees, the pylon is deflected aft two degrees, and the blade coning is zero degrees. For blade coning to be zero, the thrust developed by the main rotor must be reduced to zero. If the rotor is aerodynamically loaded and is developing one gravitational force (G) of thrust, the blade clears the tail boom by nine inches.

1.16.8.3 Results of the Computer Simulation

The simulation using the C81 program is divided into the entry and recovery phases of the simulated aircraft manoeuvre.

In the entry phase, two conditions were used. For both conditions, the aircraft was rolled into a left climbing turn at 80 knots. After stabilizing the climbing turn, a dual-inverter failure was simulated by commanding the SCAS actuators to return to the neutral position and by locking the FBW elevator in a plus eight-degree angle of incidence position.
A FBW system failure or shutdown would freeze the elevator in its last position (refer to paragraph 1.6.1); it was calculated that, given the in-flight conditions of C-GVZO at the time of the accident, the FBW elevator would have been in the plus eight-degree position.

In the first condition investigated, the pilot was allowed to instinctively apply right cyclic to overcome the left roll acceleration created by the centering of the lateral SCAS actuator (refer to 1.6.2). In the second condition evaluated, no right cyclic input was applied. It was seen that, if corrective lateral cyclic input was not made as the inverters were failed, more left roll occurred during the manoeuvre entry.

In the recovery phase, various recovery techniques from these conditions were initiated as the nose of the helicopter approached 15 degrees nose-up. Collective control reduction was initiated as the rate of climb approached 900 feet per minute.

In the recovery phase, for the first condition where the timely, instinctive right cyclic input was made, even the most severe control motions indicated that insufficient main rotor flapping would be developed to cause a tail boom strike. However, the second condition, which did not incorporate the cyclic corrective input, demonstrated that sufficient flapping could be generated to cause a tail boom strike.

The computer model showed that, if a delayed recovery using a fully aft cyclic control input was begun at 8.8 seconds from the initiation of the AC power failure, a blade flapping angle of 15 degrees was generated and resulted in tail boom contact. In a recovery using both cyclic and collective controls, the peak flapping angle of the main rotor would be only 7.5 degrees, half that required for a tail boom strike to occur.

The delay in initiating aft cyclic recovery input was intended to cover the possibility that crew attention was focused inside the cockpit for at least 4.7 seconds following the corrective action to the simulated inverter malfunction. Such a delay caused the computer model to assume an unusual attitude with a high rate of descent and at a low G. The computer model reached an attitude of 60 degrees left bank and 15 degrees nose-down, with a 2,500-foot-per-minute vertical rate of descent. When the helicopter pitched nose-down, a low-G situation developed and caused the rotor system to unload to nearly zero thrust, thereby producing a main rotor coning angle approaching zero.

A recovery effected by the use of the cyclic only produced a flapping angle of 10.5 degrees, still less than that required for a tail boom strike.

The computer evaluation showed that the combination of a delay followed by a rapid, fully aft cyclic recovery, caused the main rotor blades to strike the tail boom.

1.16.9 Warning and Caution Panel Lights and Switch Evaluation

The CASB Engineering Laboratory examined the warning and caution panel lights and determined that both the number one and the number two inverter (INV 1 and INV 2) warning lights were on when the aircraft struck the ground. The fact that these lights were illuminated indicates that both inverters were not powered at the time of the accident.

It was determined that the number two inverter switch had been selected to the OFF position. The number one inverter switch was badly damaged, and it was not possible to conclusively determine its position prior to impact.

The following warning segments were also illuminated at the moment of impact:

CAUTION PANEL

ANTI-ICE ON : Engine 1 and Engine 2
OIL : Engine 1 and Engine 2
FUEL PRES : Engine 1
OIL COOLER
BLOWER OFF
INV 1
INV 2

PILOT'S ANNUNCIATOR PANEL

MASTER CAUTION
#1 ENG OUT
#2 ENG OUT
ROTOR RPM

CO-PILOT'S ANNUNCIATOR PANEL

MASTER CAUTION
#2 ENG OUT
ROTOR RPM

The remaining warning segments were examined, but it could not be determined whether they were on at impact.

1.16.10 Main Rotor Blade Examination

The red main rotor blade was the more severely damaged of the two blades. It was reconstructed and examined to ascertain if there was any evidence of blade delamination. There were several kinds of skin separation evident during these examinations, as well as two very small areas of reduced bonding strength, and areas that were void of bonding material. These bonding voids were small and considered to have had no effect on blade integrity. No discrepancies beyond qualified manufacturing limits were found in the blades, and a review of the blades' manufacturing records showed normal processing.

One large piece of upper blade skin had separated from the trailing edge strip and noneycomb core, extending from its aft edge (coincident with the trailing edge of the blade) to near the aft edge of the spar. The separation occurred between the skin and the bonding adhesive. Forward of this area, the failure changed to core failure and then skin failure at the aft edge of the spar. The skin had not delaminated from the spar and had blue paint transfer near the forward edge and probable paint transfer near the aft edge. The core pieces generally showed intact adhesive with strands of fibreglass from the skin adhering to it. Microscopic examination showed printing of the fibreglass strands as well as fibreglass in the adhesive layer, indicating that good contact with the skin had been achieved in the bonding process. One area showed slightly glossier adhesive with less fibreglass adhering to it but still had evidence of contact during the blade bonding.

The upper skin was broken up into numerous pieces, showing mixed failure modes, including large areas of delamination between the fibreglass tape plies. The lower skin was also broken up, and several pieces showed scratch continuity marks and fuselage contact marks.

There was a permanent overtwist in the tension-torsion strap pack for the red blade that indicated a severe leading-edge-down rotation. Similarly, there was evidence of overrotation seen in the strap pack of the white blade. Additionally, there were deep score marks inside the red blade grip that showed a 150- degree leading-edge-down rotation, coinciding with the twist in the tension-torsion strap pack.

The CASB Engineering Laboratory examination concluded that failure of the afterbody skin and the subsequent breakup of the red blade was the result of massive impact and severe torsional forces experienced during contact with the fuselage and tail boom.

The white blade was much less damaged than the red blade and mainly intact with leading edge and tip cap strike damage, and cracks and delaminations in the skin area. Test sections were cut to confirm the bonding in one area of the blade. The chordwise skin failure on this blade appeared to be caused by the compressive loading of the blade afterbody.

During the analysis of the blades, accurate drawings of the red blade breakup patterns for the upper and lower surfaces were superimposed on each other. This was done by aligning the trailing edges and trim tabs of the drawings placed back to back and then illuminating them on a light table, so that both sides of the breaks could be seen simultaneously. It was apparent from the oppositely directed 45-degree breaks in the upper and lower skins that the blade fracture pattern showed torsional failure consistent with the red blade having been severely twisted leading-edge-up. That the red blade actually rotated leading-edge-up to an extreme angle was shown by the swirl mark on the mast caused by the pitch horn weight assembly located on the hub of the red blade.

Additionally, during the examination of the white blade, the piece of red blade, including the outboard part of its trailing edge strip, trim tab, some skin, and some core, was placed over the white blade, aligning the trailing edges and trim tabs. The spanwise location of the afterbody bridge failure and the trailing edge strip breaks were at very nearly the same blade stations. This similarity of breaks in the skin and trailing edges indicated similar structural integrity at the time of main rotor blade contact with the tail boom.

Although flaws were noted during examination of the blade sections, no evidence was found to indicate that spontaneous blade delamination had occurred as the initiating event.

1.17 Additional Information

1.17.1 Mast Bumping*

In conventional rotor systems, the blades are allowed to flap vertically to allow for the various in-flight aerodynamic forces experienced during rotation. The flapping angle is generally restrained to specific limits to ensure safe start-up and shutdown of the rotor. These restraints are usually ensured by flap restrainers or droop stops; however, in the teetering-type rotor system, a far simpler device is employed. In this case, the downward and upward flapping is limited by a static stop on the blade yoke contacting the surface of the mast itself.

In normal flight, the flapping angles are quite small, perhaps two or three degrees, which represents about 15 per cent of the maximum designed flapping allowable. These angles increase with higher forward speeds, gross weights, density altitudes, lower rotor rpm, and turbulence. Manoeuvring at low speeds and/or sideslips can induce even larger flapping angles, up to 60 to 70 per cent of the maximum allowable.

When the flapping angles begin to reach the system limits, blade yoke-to-mast contact is imminent. Exceeding the flapping limit produces yoke-to-mast contact, and this is the beginning of 'mast bumping'. Fully developed mast bumping is usually catastrophic because the mast is either crushed or bent by the violent contact of the yoke on both sides of the mast which then separates at the crush points. The Bell 214ST has a strengthened mast that does not crush.

Mast bumping, therefore, is a result of excessive main rotor flapping, which is usually directly related to inappropriate pilot response to low-G manoeuvres. In these circumstances, if the pilot controls the blade flapping angles adequately, then the helicopter will not experience mast bumping. Typically, the combination of forward cyclic and reduced collective lever produces a low-G situation, and these actions unload the main rotor, essentially cancelling the rotor lift, and therefore rotor thrust. The absence of main rotor thrust makes the lateral cyclic control ineffective, and hence, the fuselage does not respond to the pilot's lateral cyclic input.

* The term 'mast bumping' is used in this report because of its common usage in the helicopter industry. It should be noted that Bell Helicopter prefers the use of the words "excessive main rotor blade flapping."

In normal flight, the main rotor on a Bell helicopter has its lift vector tilted slightly to the left to produce a left lateral force component. This component offsets the right lateral thrust of the tail rotor which is counteracting the main rotor torque. When blade pitch is lowered and the cyclic pushed forward to the point where the rotor is producing appreciably less than 1G of lift, the tail rotor thrust, which is still being applied to the right and now acts above the fuselage centre of gravity, no longer has the offset thrust from the main rotor to counteract rolling moments to the right. Consequently, a rolling moment is established, and the fuselage rolls to the right beneath the rotor system. The fuselage, in its action of rolling to the right, is actually swinging to the left beneath the rotor disk, simulating the response of the opposite cyclic input, that is, to the left.

Pilot lateral cyclic control input to the left to counteract actual fuselage roll has no effect on the fuselage attitude. Actually, the situation is worsened because the left cyclic input moves the rotor disc down on the left side due to the aerodynamic reaction to the blade angle change, thereby further decreasing the clearance between the blade yoke and the mast. Correct recovery technique is smooth application of aft cyclic to restore rotor thrust, followed by lateral cyclic to level the helicopter.

Depending on the severity of the manoeuvre, in a teetering-type rotor system, lateral cyclic input can lead to contact between the mast and the flapping stops of the rotor, to mast bumping, and catastrophic damage. These conditions would likely be amplified if the helicopter were in a left turn, because the weight moment of the fuselage would be added to the right rolling moment created by the tail rotor, causing faster movement of the mast into the yoke.

1.17.2 Aerodynamic Analysis of Collective Sleeve Failure

A tension failure of the collective sleeve under medium power settings would result in the upper portion of the broken collective sleeve moving upward on the mast as a result of the aerodynamic loading on the blades. The main rotor blades would abruptly decrease their pitch setting, offload the rotor system, and achieve a new low blade angle. The rotor blades would stabilize at an angle of +8.35 degrees at the root or -0.03 degrees at the blade tip. This blade angle equates to 26 per cent of the collective control position. It is likely that the blades would overshoot this equilibrium angle because of the abruptness of the failure; the actual amount of overshoot would depend on the blade angle at the time of failure and the amount of friction provided by the collective friction collet.

1.17.3 Test Flights Conducted by the Operator

During both VFR and simulated IFR test flights conducted in Churchill, the aircraft was flown in a left climbing turn to approximate what is believed to have been the climb-out attitude of the helicopter involved in the Argentia accident prior to its departure from controlled flight.

On one occasion, the pilots conducting the Churchill test flights flew C-GVZO in hazy sky conditions with little or no horizon reference. They reported some difficulty in establishing full control under these conditions when artificial delays and abnormal tail rotor control inputs were introduced. The delays were incorporated into the test flights in order to experience the helicopter's attitude excursions and the consequent recovery requirements, assuming that the Argentia pilots delayed their recovery actions for undetermined reasons.

A total AC failure during a left climbing turn was simulated by turning off the number one and number two inverters simultaneously. When AC power was lost, the aircraft, with little or no accompanying roll, smoothly and quickly pitched up 10 to 14 degrees within three to four seconds. This resulted in a 1,500-foot-per-minute rate of climb within the same time period. The tendency of the pilot was to lower collective and prevent altitude gain when he noticed either the AC power failure or the attitude change. The nose continued to pitch up until the collective was lowered. Abruptly lowering the collective changed the attitude; the aircraft pitched down and yawed left, thereby increasing the left bank. The pilots stated that, had corrective action not been initiated within two to three seconds, the pitch-down attitude, left bank, and yaw would have become excessive; also, the degree and abruptness of the aircraft's attitude change increased with the speed and the rate and amount of collective movement.

The Bell 214ST has no mechanical collective to yaw coupling, and, with no power to the AFCS, the aircraft will yaw left when the collective is lowered quickly. It was noted during the Churchill test flights that some pilots, accustomed to aircraft that have mechanical collective to yaw coupling, were slow to input right pedal to correct for the left yaw when the collective was lowered. Consequently, in order to simulate this reduced input from such pilots, the tail rotor control inputs were lessened by the pilots conducting the test flights.

1.17.4 Further Test Flights

Following the completion of the Churchill test flights, the operator, in consultation with a private aircraft accident investigation consultant, decided that, for the purpose of precise documentation, further test flights were required. The consultant coordinated with the operator's senior flight personnel and drew up a series of test sequences involving left and right climbing turns with either the number one inverter off, the number two inverter off, or both inverters off. These flights were to be conducted at 70 to 80 knots indicated airspeed (IAS) at a climbing vertical speed indication (VSI) of 500 feet per minute. Following the simulated inverter malfunctions, the pilot in the left seat was to record those components, sensor inputs, or systems with respect to the AFCS that discontinued operation. This pilot was also required to identify whether or not those components and systems remained disabled and, in the case of a single-inverter failure, if they automatically switched to the remaining inverter. He was also to identify any item that required manual reset by the crew before it would remain in operation. In addition, he was to record any movement in the attitude display indicator (ADI) and note any significant attitude reaction by the aircraft.

Documents recovered at the accident site included five of eight pages written by the pilot-in-command and a torn piece of the test-flight schedule prepared by the Okanagan accident consultant. The overall test-flight schedule comprised eight separate tests which systematically involved climbing turns to the right and left, followed by failure of each inverter, both individually and simultaneously. By interpretation of the five sheets recovered, the first and second tests were single-inverter failures in a left turn. The third and fourth tests were as the first two, but in a right turn. The fifth and sixth tests were dual-inverter failures in a left turn, while the last two tests were to the right. Tests one, four, and five were not found; however, tests two and three were completed by the crew, whereas tests six, seven, and eight were not.

1.17.5 Operational Control Aspects of the Flight

Okanagan did not feel it necessary to seek assistance from Bell or Transport Canada in the preparation and conduct of the test flights in Churchill or Edmonton, nor are there specific regulations requiring an operator to notify or consult with either Transport Canada or the aircraft manufacturer prior to conducting experimental or test flights.

1.17.6 Characteristics of C-GVZO

The mast was changed because the collective collet friction wear sleeve was worn beyond the allowable limits. The number of times the collective collet set was changed was greater than the rest of the fleet. The collet set was found broken three times during aircraft service and also during post-accident inspection. The in-flight rotor tracking system on this aircraft had been disconnected and finally removed. The left-hand-side nodal beam conical torsion mounts had been replaced at 1,690.7 airframe hours. Okanagan Helicopters was monitoring a fatigue crack in the right nodal beam mounting system; the fatigue crack did not extend during the aerial breakup. Bell was aware of the fatigue crack and had approved continued flight.

Shortly after the company took delivery of C-GVZO, it was noted during training that the aircraft reacted with an abrupt attitude change accompanied by a heavy vibration during entry into autorotation at speeds approaching the one-engine-inoperative, never-exceed speed, (Vne_{oei}), of 120 knots. However, during initial FAA certification test flights, it was observed that the Bell test helicopter handled acceptably in all phases of autorotation. The benign nature of the Bell 214ST type in autorotation was further demonstrated to CASB personnel during a flight at the Bell flight facility in Fort Worth, Texas.

1.17.7 Witnesses

Several witnesses saw the helicopter before, during, and after the aerial breakup. Witnesses observed the helicopter in a left turn and the main rotor striking the tail boom. One witness reported that the rotor had stopped after it separated from the fuselage.

1.17.8 Technical Report on the Study of Selected Aspects of the Bell 214ST Certification

In the light of the two Bell 214ST accidents in Canada, the CASB examined certain aspects of the Bell 214ST certification process. To facilitate this examination and to assist with the interpretation of pertinent certification documentation from Bell, the FAA, and Transport Canada, meetings were held with airworthiness and certification authorities from Transport Canada in Ottawa and the FAA in Fort Worth, Texas. In addition, detailed certification data and information on the automated flight control and electrical systems were provided by the manufacturer and were carefully examined. Finally, a flight crewed by a Bell test

pilot and CASB investigators was undertaken to achieve a better appreciation of the flight characteristics of this helicopter and the functioning of its control and electrical systems. The specific areas of interest were aircraft stability, autorotation characteristics, AFCS operation in normal and abnormal conditions, electrical systems, standby attitude indicator, IFR climb performance, and certification approval.

The certification was examined to establish if the Bell 214ST met the certification criteria. This examination established that, for the certification elements under consideration, the Bell 214ST met the certification criteria in all respects.

2.0 ANALYSIS

2.1 Introduction

The investigation determined that the main rotor blades struck the tail boom during in-flight manoeuvring, resulting in the breakup and destruction of the helicopter. It is not possible for the blades to strike the tail boom of an intact aircraft without yoke-to-mast contact occurring first. To determine the initiating event that led to the yoke-to-mast contact and the resulting in-flight breakup, the investigation analysed the potential for blade strikes in relation to the following possibilities:

1) Collective sleeve failure;
2) Pilot control inputs;
3) Collective friction collet binding; and
4) Main rotor blade delamination.

The investigation also examined operational considerations regarding the conduct of the flight, selected aspects of the certification of the helicopter, the unusual flight characteristics of C-GVZO, and the collective sleeve manufacturing process.

2.2 Blade Strike Analysis

It is important to recognize that the whole elastic mast-bending and blade strike sequence is of an extremely short duration, approximately 20 blade passages or two seconds at the normal main rotor speed of 287 rpm. Of these 20 potential blade strikes, only seven have been accounted for.

It was determined that, with reference to the fuselage, the main rotor disc was tilted left and aft immediately prior to the first blade strikes. The blade path was determined to have been in an initially flat and then an upward motion as the blades passed over and through the tail boom. Both these determinations are supported by the tail boom impact lines and evaluations of the blade angles.

Given the evidence available, it was impossible to accurately determine the actual blade impact sequence. There was, however, some evidence that permitted the identification of individual blades with specific tail boom contacts. Therefore, the following discussion compares the blade damage to the fuselage damage and, where possible, identifies matching contact points.

Although the complete blade strike sequence could not be determined, strikes 'A' and 'B' were found to be the initial two contacts with the tail boom, followed by strikes 'C' to 'G' in an unknown order.

The first two blade strikes were attributed to the red and white blades and were designated 'A' and 'B'. The blade tip angle upon tail boom contact was not precisely determined; however, there was no evidence of tail boom contact on the outer trailing edge of either blade tip, indicating that the blade tips were at a negative pitch angle. This blade tip angle has been calculated by geometric analysis to have been at least -12 degrees. The two tip strikes were quite close on the drive shaft cover, indicating that both blades were under similar collective pitch control forces, with both pitch links still intact. The crushing of the cover indicated both vertical blade impact forces and high speed horizontal blade impact forces consistent with the tip path plane being tilted aft over the tail boom. This aft tip path plane suggested that the tail boom area was the lowest sector of rotor disc flapping which was consistent with either aft cyclic application, aft mast bumping, or both.

It was therefore concluded that, upon contact with the tail boom, blade strikes 'A' and 'B' were at a negative pitch angle consistent with either a low collective lever position or a collective sleeve failure.

After the two initial tail boom contacts, the amplifying elastic mast-bending would have begun to influence the blade tip path plane. The blade strikes, 'E', 'F', and 'G', are thought to have occurred during the upward departure of the main transmission because of the increasing angles of tail boom contact.

The mast first contacted the soft stops on the blade yoke and subsequently the yoke itself. This is witnessed by the contact marks on the mast. The indentations on the mast caused by the red blade are deeper, indicating higher red blade impact forces. For the blade yoke to have experienced such contact with the mast, mast bumping must have been established. The mast would have been bending elastically simultaneously with the mast bumping, and the situation would have quickly developed where the blades were flexing down towards the tail boom, ultimately contacting the drive shaft cover, thus precipitating the tail boom destruction.

The deep contact mark on the white blade leading edge mated with the damage at tail boom station 243.7, and the equally distributed contact marks on the blade indicated a blade angle upon contact of nearly zero, with an intact white pitch link.

As the white blade passed through the tail boom and drive shaft, it was also rising relative to the tail boom. It was determined that the white blade began to rotate leading-edge-down as it contacted the drive shaft, which was witnessed by a permanent twist to the white blade tension strap pack. Such a rotation would apply a large tensile force on the pitch link, and it was therefore concluded that the pitch link failed in tensile overload caused by the blade twisting. Upon release from the tail boom, the white blade rotated leading-edge-up as the strap pack unwound, causing even more deviation in the rotor blade disc.

Examination of the red blade also revealed that the twist in the tension-torsion strap pack and the deep score marks in the blade grip indicated a 150-degree leading-edge-down rotation. This twisting force was likely to have occurred when the red blade contacted the tail boom and drive shaft with a negative blade tip angle. As the red blade struck the tail boom, it would have begun to twist leading-edge-down. Upon drive shaft contact, the blade would have twisted more severely, causing overrotation of the strap pack and the deep scoring of the blade grip. Additionally, the red blade twisting would have been mostly responsible for the beginning of the blade breakup.

As the red blade passed through the drive shaft and tail boom, the tension strap pack unwound rapidly and rotated the blade back past its previous position to a very high angle of attack, which was confirmed by the contact mark on the mast made by the red blade pitch horn.

With such excessive blade angles, demonstrated by evidence in both blade grips and retention strap packs, the rotor disc would have become erratic and no longer controllable. This aerodynamically unstable situation degenerated rapidly and, combined with the tail boom impact forces, would have finally resulted in the departure of the transmission and the breakup of the machine. The rapid axial rotation of the blades would also have been a significant contribution to main rotor rpm decay.

2.3 Collective Sleeve Failure

This particular collective sleeve had been in use for over 2,000 hours and had come from a batch of sleeves that had been improperly manufactured. This improper process resulted in the presence of HEC, and, as a result, the sleeve was weakened.

After the previous collective sleeve failure in Oman in February 1984, an AD was issued requiring FMPI of all collective sleeves from this particular batch. Although the sleeve from C-GVZO had about 1,197 hours at the time the AD was issued and accordingly was required to be inspected at 2,500 hours, the operator contracted to have the sleeve inspected at this earlier flight time. This inspection was deficient with respect to the surface preparation of the sleeve. It was later determined that the sleeve was cracked at the time of this inspection, but detection was prevented by the presence of paint and primer on the surface.

It was determined that a number of HEC flaws had been present in the sleeve from the time of manufacture. It was also determined that this HEC had not developed further during the component's service life. The extension of some of the cracks by fatigue was considered insufficient to cause collective sleeve failure under limit in-flight load.

The fracture of the sleeve showed the characteristics of failure caused by high tensile loading, that is, tension overload. No evidence was seen of bending stresses on the fracture surfaces.

Bell's pre-certification test-flight program calculated the stresses experienced by the sleeve under limit in-flight load. Test data were gathered regarding its ability to withstand loads resulting from manoeuvres within the normal flight envelope.

Calculations showed that the strength of the component would not have been reduced to the point of fracture at limit in-flight loading. The slow fatigue propagation from the HEC in the sleeve over its operational life supports the observation that the failure occurred from a transient load well in excess of those previously experienced.

Since the introduction of C-GVZO into Canada in October 1982, the helicopter had been involved in both company pilot training and regular offshore support for the oil industry in Canada and Alaska. There were at least 10 initial pilot type-endorsements on this aircraft, each of which would likely have exposed the helicopter to

the unusual autorotation entry vibration. The aircraft had been primarily operated in the IFR offshore environment and had been loaded often to its maximum all-up weight (MAUW) of 17,500 pounds. The aircraft had been exposed to at least moderate turbulence and high weights during its service life. The estimated in-flight weight of the aircraft at the time of the accident was less than 14,500 pounds, that is, at least 3,000 pounds less than MAUW. In consideration of the rate of propagation of the cracks, it is unlikely that, in this accident, the collective sleeve would have experienced loads in excess of those previously seen, unless the helicopter was flown outside the normal operating parameters.

There were several series of score marks on the main rotor mast which suggested that collective sleeve failure occurred during the elastic mast-bending caused by mast bumping. There was also evidence to show that some of these marks were made by the internal edge of the fracture. Consequently, these particular marks could only have been made after the collective sleeve had fractured.

It would not be possible for the fracture surface of the upper portion to leave circumferential marks on the mast, unless it contacted the lower part. That the collective sleeve separated, allowing the upper part to travel up the mast and return to strike the lower part of the broken sleeve, is evidenced by the two projections found on the internal surface of the fracture and scrape marks on the external surface of the upper part of the sleeve which match with the fracture edge of the lower part. These two projections were shown to be responsible for some of the upper area witness marks on the mast. It was concluded that the marks were made by both the lower fracture edge and the upper fracture edge containing the two projections formed when the two parts made contact.

The score marks on the mast showed that elastic mast-bending occurred with the collective control initially at the 50 to 60 per cent position. The score marks also suggested that the collective sleeve failed in about the 40 per cent collective control position. The upper edge of the marks equated to a collective control position of less than zero, which was not possible with an intact collective sleeve.

Had the initial tail rotor drive shaft cover contacts occurred with the collective sleeve intact and the collective control in the 50 to 60 per cent position, evidence of a slightly positive blade tip angle would have been expected. However, the initial strikes

display evidence of a definitely negative blade tip angle. This blade tip angle was consistent with a low collective control position or a collective sleeve failure. As there was some evidence to show that the collective control was at the higher position of 40 per cent, it was concluded that the collective sleeve failed. It was therefore likely that, as a result of the mast-bumping and mast-bending forces, the weakened sleeve had been subjected to a transient load that exceeded its material strength. The sleeve then failed and caused an immediate offloading of the main rotor blades, which would have probably overshot the calculated stabilized blade tip angle of -0.03 degrees before returning to this angle. During these large excursions of blade angles, the blade path would have been erratic with the blades striking the tail boom, causing their bending, twisting and breakup. Simultaneously, the mast was bent permanently, the nodal beam mounts failed, and the transmission departed the helicopter.

2.4 Control Inputs

2.4.1 C-81 Computer Simulation

The C81 helicopter computer analysis simulated a collective sleeve failure or a rapid down collective input by the pilot. The simulation showed that blade to tail boom contact could occur if the aircraft attitude was allowed to diverge to unusual limits and, following a delay of three or four seconds, recovery was initiated with rapid, full-aft cyclic input. Such a delayed input would have precipitated mast bumping which would have quickly developed into blade to tail boom contact.

The C81 program did not take into account the probable blade angle overshoot past the stabilized blade angle calculated for collective sleeve failure. Otherwise, the program approximated C-GVZO's reactions to the inverter failures, which was shown by the test-flight reports from the Churchill test flights. The simulation demonstrated that the helicopter could have been placed in a tail boom contact situation following certain pilot control movements after either a collective sleeve failure or a rapid down collective selection.

2.4.2 In-flight Inputs

In normal flight, the main rotor blade flapping angles are small; as the flapping angles increase, the clearance between the main rotor mast and the blade yoke decreases, and the possibility of mast bumping becomes greater. Low or negative G forces worsen the flapping angles, increasing the possibility of yoke-to-mast contact. Mast bumping would not necessarily cause tail boom contact; however, tail boom contact requires yoke-to-mast contact.

Mast bumping is probable when the rotor disc is commanded to move at a rate faster than the fuselage can follow. Recovery using rapid cyclic movement would be expected following the sudden recognition of unusual or unexpected fuselage attitudes; such attitudes were observed during the test-flight sequences previously conducted at Churchill. Furthermore, the introduction of intentionally delayed reaction times would aggravate the fuselage response.

Without the benefit of flight recorder information, it was not possible to determine the exact flight profile of the accident flight. It was known from the test-flight schedule that the pilots were planning further inverter-off testing during climbing turns. The helicopter was observed in a left turn just prior to the in-flight breakup. The light bulb and switch analyses indicate that both inverters were off at the time of the accident. Consequently, in view of the pilots' test-flight sheets recovered at the accident site, it is most likely that the crew was either in the final stages of completing the first dual-inverter failure test, test number five, or beginning the second dual-inverter failure test, test number six. These tests involved investigating the effects on a climbing left turn at 80 knots after turning off both inverters.

During a left, climbing turn with both the SCAS and AARS inoperative as a result of both inverters being selected off and without pilot input, the aircraft's left roll attitude will not change significantly and the pitch attitude will increase in the nose-up direction. Upon lowering the collective lever, the roll attitude will increase to the left and the pitch attitude will decrease to a nose-down position. This reaction of the aircraft would place it in a low-G situation, thus exposing it to the possibility of mast bumping.

A collective sleeve failure in a left turn would also be conducive to mast bumping. However, in this situation, for the weakened sleeve to have failed, a

transient overload would have been required. It has been determined that there was elastic mast-bending with the collective control in the 50 to 60 per cent position, prior to collective sleeve failure. Since bending of the mast would require yoke-to-mast contact, it is concluded that the collective sleeve failed after mast bumping began.

2.5 Characteristics of C-GVZO

The continuing in-flight vibrations, the excessive mast wear, the frequent collective collet replacement, the replacement of part of the nodal beam structure, the fatigue crack in the right nodal beam mounting system, the disconnection and removal of the rotor tracking system, and the unusual autorotation entry characteristics collectively suggest that this aircraft was not typical. The origin of these difficulties was not determined, nor was any evidence found to link them with the accident.

2.6 Collective Friction Collet

The collective friction collet failed in high-cycle fatigue due to excessive vibration forces transmitted from the rotor within the previous 185 hours. The history of collet replacements and failures supported the presence of this pre-existing vibration and was indicative that the vibration forces had been present since the helicopter first entered service.

The several regular grooves in the mast friction wear sleeve demonstrated an excessive wear rate, considering the component service life of 380 hours; however, the wear pattern was considered normal. Consideration was given to the possibility that the additional rub marks on the friction sleeve wear were evidence of binding of the friction collet on the mast surface, leading to an overload failure of the collective sleeve. Because increasing the collective pitch creates a tension force within the collective sleeve, jamming of the friction collet would have enabled an increased tension force from the collective actuator to be experienced by the collective sleeve.

It has been calculated that the collective servo did not have the required power to transmit the overload forces necessary to fracture the collective sleeve in its weakened condition.

The actual wear on the friction wear sleeve was considered excessive but not in a manner which could have prevented the full movement of the friction

collet over its normal range of travel. The fractured pieces were contained within the attachment fittings and would not have been able to misalign and create restriction or binding. The worst effect of the one significant fracture would have been a minor reduction of the friction applied by the collet assembly, which could have allowed any rotor-generated vibration to become more noticeable by the occupants in flight. This reduction in the vibration damping friction was a possible explanation for the previously reported in-flight vibration. It was concluded therefore that the fractured friction collet did not contribute to this accident.

Although the collective friction collet is not considered causal in this accident, the significance of the fatigue failure, the excessive wear rate, and the origin of the abnormal vibration was not determined.

2.7 Main Rotor Blade Delamination

The manufacturer conducted a detailed series of calculations concerning the in-flight forces and loads experienced by the Bell 214ST rotor blades. In consideration of the suspected pre-breakup flight manoeuvres, the calculations demonstrated that the aerodynamic forces acting on the surfaces of the blades were insufficient to cause the spontaneous delamination of blades built to manufacturer's specifications. The CASB Engineering Laboratory, in concert with the manufacturer, conducted an examination of the blade pieces and determined that severe torsional deformation of the blades caused several types of skin separation, delamination, and blade core fracture. Additionally, it was determined by matching blade and fuselage contact marks that large upper blade surface delamination took place upon impact with the engine cowling.

Overall, the blade breakup is consistent with severe twisting and impact damage caused by tail boom contact. The main rotor blades were carefully examined, and it was seen that most of the blade damage was caused by impact with the tail boom. This was supported by matching blade and fuselage contact marks, some of which required blade skin integrity upon impact. There were some blade-skin bonding irregularities observed; however, as these were small and within manufacturing specifications, it is considered that they did not contribute to the blade breakup. Therefore, it was concluded that the blades skins were intact upon initial contact with the tail rotor drive shaft cover on the tail boom.

2.8 Operational Considerations

In order to analyse the operational aspects of this accident, the conduct of the flight was examined.

Okanagan Helicopters believed that the helicopter reacted in an unusual and unexpected manner following a simulated dual-inverter failure. This belief was based upon the in-house reports of company pilots who had systematically experimented with C-GVZO, in both VFR and simulated IFR flight, by turning off inverters while the aircraft was established in climbing left- and right-hand turns. These test flights took place because of the operator's opinion that the Argentia accident involving C-GSYB suggested an inverter-related irregularity.

The pilots who had originally conducted these test flights observed that excessive, unusual aircraft attitudes would result if pilot corrective action was not taken or was delayed, especially during IFR flight. It was their opinion, however, that these attitude excursions were within the aircraft certificated flight envelope and that the tests were benign.

Both pilots had low experience levels on the Bell 214ST helicopter; however, they were highly experienced helicopter pilots, with over 1,000 hours each on similar Bell helicopters, the Bell 212 and Bell 214B. While the nature of the simulated inverter failures was considered by the Okanagan pilots to be no more rigorous than an IFR training flight, it was experimental; the actual conduct of the test flight and the procedures being employed to diagnose and document the aircraft systems and responses were not typical of a training flight.

If the pilots delayed recovery action in order to assess the effects of double-inverter failure and thereby inadvertently allowed the aircraft to exceed the certificated flight envelope, any rapid cyclic corrective action would have increased the possibility of yoke-to-mast contact. The pilots in the Edmonton accident flight were personally involved in the previous Churchill test flights which incorporated experimentation with various time delays during the test-flight sequences. During the Edmonton tests, there would in all probability have been delays in initiating corrective action, while the pilots assessed and observed systems and aircraft response. The exact nature and duration of any time delays experienced by the Edmonton pilots could not be determined.

The Bell 214ST is a sophisticated and high performance helicopter. The result of shutting down the AC electrical system was suspected as being the cause of a previous fatality. The aircraft reactions and the recovery techniques necessary for safe flight for all scenarios were unknown at the time of the test flights and were therefore experimental in nature.

Had the crew been failing basic essential aircraft systems that produced degraded flight handling qualities and, at the same time, delaying control corrections, the crew would have been operating the aircraft outside the normal certification standards and flight manual limitations and procedures.

Okanagan did not feel it necessary to seek assistance from Bell or Transport Canada in the preparation and conduct of the test-flights in Churchill or Edmonton, nor were there any requirements to do so. There are no specific regulations with respect to the conduct of experimental test flights requiring an operator to notify or consult with either Transport Canada or the aircraft manufacturer.

However, there are regulations and procedures that should cause the operator to contact Transport Canada or the aircraft manufacturer. The Certificate of Airworthiness (C of A) requires, among other things, that the aircraft be operated in accordance with its approved flight manual.

2.9 Certification Aspects of the Bell 214ST

Following the two major occurrences involving the Bell 214 ST - Argentia 85-H50002 and Edmonton 85-H50007 - a study concerning selected aspects of the helicopter certification was conducted by the CASB. This in-depth study concluded that, for those areas of specific interest, the Bell 214ST met the certification criteria in all respects. In particular, the helicopter demonstrated acceptable control characteristics when exposed to single- and double-inverter power failures. The flight conducted by the CASB investigation team showed that the helicopter did not exhibit any abnormal or even difficult handling qualities when the AC inverters were selected off and the approved contingency procedures in the RFM employed. Aircraft control and response were observed to be affected by the AC power loss; however, no degraded or unexpected control inputs were required to keep the helicopter under control, when timely control inputs were made.

2.10 Flight Recorders

The investigation into the causes and factors that led to this accident was hampered by the lack of accurate information which would have been provided by FDR/CVR equipment. In the absence of such information, the Board had to use other, less reliable and more time-consuming methods in an effort to determine the sequence of events leading up to the accident.

2.11 Summary

Okanagan Helicopters, with their consultant, participated in the preparation of a test-flight schedule for C-GVZ0, in an attempt to confirm Okanagan's suppositions concerning the operation of the inverter system.

The helicopter was determined to have inadvertently entered mast bumping, thus resulting in elastic mast-bending to the point where the blades came into contact with the tail boom. The extreme forces of tail boom contact bent the mast permanently, fractured the pitch links and nodal beam mounts, severed the tail boom assembly, and then continued to destroy the aircraft. The helicopter was observed by several witnesses to break up in flight. The breakup sequence was estimated to have occurred in approximately 10 main rotor revolutions or in about two seconds.

The following scenarios were considered:

1) Collective sleeve failure;
2) Pilot control inputs;
3) Collective friction collet binding; and
4) Main rotor blade delamination.

The evidence did not support the scenarios of spontaneous blade delamination or collective friction collet binding.

Although the severity of the mast bumping in itself may have led to the severing of the tail boom, the failure of the collective sleeve may also have contributed to the rotor blade to tail boom contact.

The collective sleeve had been improperly manufactured and, as a result, was weakened by hydrogen embrittlement cracking. The sleeve failed as a result

of a loading which exceeded the strength of the cracked sleeve and was of a value higher than those previously experienced. This load was caused by the forces experienced during the elastic mast-bending resulting from mast bumping. Mast bumping occurred most likely as a result of control inputs during manoeuvring flight.

3.0 CONCLUSIONS

3.1 Findings

1. Okanagan Helicopters undertook a series of test flights in an attempt to obtain information about the causes of the earlier Bell 214ST accident in Argentia.

2. During in-flight manoeuvring, mast bumping occurred which induced elastic mast-bending.

3. The elastic mast-bending created a load on the collective sleeve which caused it to fail.

4. The main rotor blades struck the tail boom, resulting in the breakup and destruction of the helicopter.

5. Although the severity of the mast bumping in itself may have led to the severing of the tail boom, the failure of the collective sleeve may also have contributed to the rotor blade to tail boom contact.

6. The collective sleeve was defective, having a large number of defects characteristic of hydrogen embrittlement cracking through approximately 28 per cent of the cross-sectional area of the sleeve.

7. The hydrogen embrittlement cracking resulted from an improper deviation from the authorized manufacturing process.

8. Bell Helicopter Textron had identified the defective batch of sleeves and the Federal Aviation Administration had issued an Airworthiness Directive requiring a fluorescent magnetic particle inspection.

9. The sleeve on C-GVZO to which the Airworthiness Directive applied had about 1,197 hours at the date of issue of the AD and was not required to be inspected until 2,500 hours.

10. Okanagan Helicopters Limited contracted to have the sleeve inspected by the fluorescent magnetic particle inspection method at the time of issue of the AD; the inspection was not in accordance with the Bell Helicopter Textron Alert Service Bulletin with respect to the removal of surface paint.

11. The sleeve was cracked at the time of the inspection, and the detection of these cracks was prevented by the incorrect method of inspection.

12. The flight loads experienced by the collective sleeve throughout its service life of 2,039 hours did not promote any significant growth of the cracks by fatigue.

13. The accident aircraft had had six collective friction collet replacements during its life, which is an abnormally high replacement rate in comparison with the Bell 214ST fleet.

14. The collective friction collet had failed because of high-cycle fatigue as a result of vibration forces.

15. The mast friction wear sleeve was excessively worn, considering the component life.

16. Previously, the mast had been changed at 1,659 airframe hours because the mast friction wear sleeve was worn beyond limits, but the source of this wear could not be determined.

17. The wear did not prevent the full movement of friction collet over its normal range of travel.

18. The collet was contained positively inside the splined drive plate assembly, and the fractured parts were not permitted to migrate so as to create restriction or binding.

19. The origin of the vibration leading to the collective friction collet fatigue failure was not established.

20. There was a reported vertical vibration which was possibly associated with the reduced effectiveness of the fractured collective friction collet.

21. The main rotor blades were intact upon initial contact with the tail boom.

22. There are no regulations or guidelines with respect to the conduct of other-than-normal test flights requiring an operator to consult with either Transport Canada or the aircraft manufacturer.

23. A study conducted by the CASB concerning selected aspects of the helicopter's certification concluded that, for those areas of special interest, the Bell 214ST met the certification criteria in all respects.

24. During entry into autorotation at speeds approaching 120 knots, the handling characteristics of the accident aircraft were atypical.

25. The aircraft was not equipped with a flight data recorder or a cockpit voice recorder, nor was either required by regulation.

26. The accident investigation into the causes and factors that led to this occurrence was severely hampered by the lack of information that a flight data recorder and a cockpit voice recorder could have provided.

27. The flight crew was certified and qualified for the flight in accordance with existing regulations.

28. The aircraft was certified, equipped, and maintained in accordance with existing regulations and approved procedures.

29. The weight and centre of gravity of the aircraft were within the prescribed limits.

3.2 Causes

Mast bumping occurred, most likely as a result of control inputs during manoeuvring flight, causing the weakened collective sleeve to fail. Although the severity of the mast bumping in itself may have led to the severing of the tail boom, the failure of the collective sleeve may also have contributed to the rotor blade to tail boom contact. The collective sleeve was cracked and weakened by an improper deviation from the authorized manufacturing process.

4.0 SAFETY ACTION

4.1 Action Taken

4.1.1 Collective Sleeve Assembly

As a result of this occurrence, on 31 October 1985, the CASB recommended that:

a. the Department of Transport identify all those Bell helicopters registered in Canada that utilize a collective sleeve manufactured under the same processes as the P/N 214-010-411-(all dash numbers) that have the serial numbers prefix A-19);

b. as a matter of urgency, the Department of Transport ensure that effective inspection procedures to detect any defects due to hydrogen embrittlement in the collective sleeves are prescribed for all those helicopters identified in para a. above, and that the inspections are carried out in full compliance with these procedures; and,

c. the Department of Transport consider removing from service and spares stock all collective sleeves P/N 210-010-411-(all dash numbers) that have serial number prefix A-19 pending reinspection. (See CASB 85-12.)

On 08 November 1985, the FAA issued Airworthiness Directive 85-23-05 which required an immediate and repetitive magnetic particle inspection of suspect collective sleeves.

Subsequently, on 20 February 1986, Bell Helicopter Textron issued an Alert Service Bulletin (ASB) 214 ST-86-33, which required the replacement of all existing Bell 214ST collective sleeves with a newly developed sleeve. The improved collective sleeve has increased shot peening coverage and is coated with tungsten carbide in lieu of chromium plating.

As of 01 November 1985, there were no helicopters operating in Canada with these defective collective sleeves, nor were there any suspect sleeves in Canadian stocks.

4.2 Action Required

4.2.1 Cockpit Voice Recorder/Flight Data Recorder Equipment

The CASB is confident that, had flight recorder information been available to the investigators, the contributing cause factors to the earlier Bell 214ST helicopter accident in Argentia, Newfoundland would have been known, thereby obviating the operator's perceived need to conduct flight tests as part of his independent investigation into the Argentia accident. Thus, this second accident may have been avoided.

Technological advances in aircraft control systems, particularly in helicopters, have significantly increased the difficulty of the task for the accident investigator. The use of CVR/FDR equipment is now more critical than ever in identifying safety problems in a timely manner, with a high degree of certainty.

The United States, Australia, and Great Britain have recognized the value of CVR/FDR equipment in accident prevention and have taken steps to adopt regulations for mandatory fitment of flight recording equipment in certain categories of helicopters. At the moment, all Canadian-registered helicopters, including those that are used or intended to be used for the carriage of passengers, are exempt from any requirement to be equipped with CVR/FDR equipment.

The CASB is aware that Transport Canada is presently formulating revisions to Canadian flight recorder regulations that will be more responsive to current accident investigation needs. On 18 November 1987, the CASB submitted a Statement of Operational Requirements (SOR) for flight recorders which included a proposal for helicopters in a weight class above 2 700 kilograms. This class of helicopter, with its significant passenger-carrying capability, is used almost exclusively in commercial or corporate operations. The revision to the order pertaining to flight recorder equipment as proposed by Transport Canada does not include provisions for the fitment of any FDR/CVR equipment in helicopters.

In view of the demonstrated need for flight recorder equipment in some helicopter operations, the CASB recommends that:

> The Department of Transport adopt the CASB's Statement of Operational Requirements for flight recorders for helicopters in a weight class above 2 700 kilograms.

4.2.2 Conduct of Test Flights

The intent of existing regulations with respect to aircraft operations is that, once certificated, the aircraft will be operated in compliance with the operating limitations specified in the approved flight manual and with the airworthiness limitations of the approved aircraft maintenance manual. These prescribed limitations apply equally during test flights; experimentation beyond these limitations at any time may be extremely hazardous.

While the operator was not prohibited from conducting his own investigation into the Argentia 214ST aircraft accident, the Board is concerned that the operator would include as part of his independent investigation a test-flight program which was experimental in nature, without prior consultation with the manufacturer or certification authorities. The Board questions the appropriateness of the operator's test-flight procedures in a number of areas. Specifically, the test flights conducted in Churchill prior to the accident included intentional pilot delays in corrective action subsequent to the introduction of a system malfunction, and experimentation was made with the tail rotor corrective control inputs. In addition, the Board questions the prudence of conducting at least one of the experimental flights in Churchill in hazy conditions which provided little or no discernible horizon.

The CASB considers that the increased risks inherent in the conduct of test flights, particularly those flights investigating the controllability of an aircraft, warrant certain precautions by the owners, operators, and pilots involved in such flights. At present, there are no general guidelines for the conduct of test flights - either for owners/operators or pilots. As a minimum, pilots conducting test flights must be guided by such principles of basic airmanship as acceptable weather and light conditions, prohibited manoeuvres, use of seat-belts and shoulder harnesses, etc. Additionally owners and operators require guidance with respect to minimum aircrew qualifications, minimum crewing, minimum operating conditions, operational controls, etc., for particular types of test flights. Therefore the CASB recommends that:

> The Department of Transport establish principles of airmanship for pilots and guidelines for owners and operators with respect to the conduct of test flights.

This report and the safety action therein has been adopted by the Chairman, K.J. Thorneycroft, and Board Members:

W. MacEachern
A. Portelance
B. Pultz
F. Thurston

Members N. Bobbitt, L. Filotas, and R. Stevenson dissented as follows:

In our opinion, it is unlikely that the accident sequence could have been initiated by deliberate control inputs during manoeuvring flight. Our reasoning is as follows:

In the absence of flight recorders, it was not possible to establish the helicopter's attitude or trajectory before it began to break up. Eyewitnesses observations suggest to us that the helicopter was in level flight with unremarkable pitch and roll angles when a marked change in noise signalled some structural failure.

Computer simulations show that, in the absence of prior mechanical failure, control inputs can cause the main rotor to contact the tail boom only under very special circumstances. These circumstances involve large pitch and roll angles that have been maintained for several seconds.

It is unlikely then that the accident sequence was initiated by the pilots' response to a simulated inverter failure during a climbing left turn similar to that presumed to have preceded the unexplained crash of a similar helicopter some months earlier.

We note that automatic flight control systems on the Bell 214ST are considerably more complex than typical on civil helicopters and that flying qualities are closely linked to their operation. Inadvertent second failures have occurred after deliberate introduction of an initial failure to the automatic control systems during training flights. Thus, we cannot rule out the possibility of an uncommanded input from the partly disabled automatic control system.

Initial failure of the weakened collective sleeve could in itself explain the mast bumping tail boom contact sequence postulated by the majority. We note that the theoretical analysis of the residual strength of the collective sleeve is based on a number of simplifying assumptions and has an undetermined margin of error.

Nor could we rule out the possibility that flow perturbations from a relatively small blade delamination might have led to a buildup of aerodynamic forces leading to the mast bumping

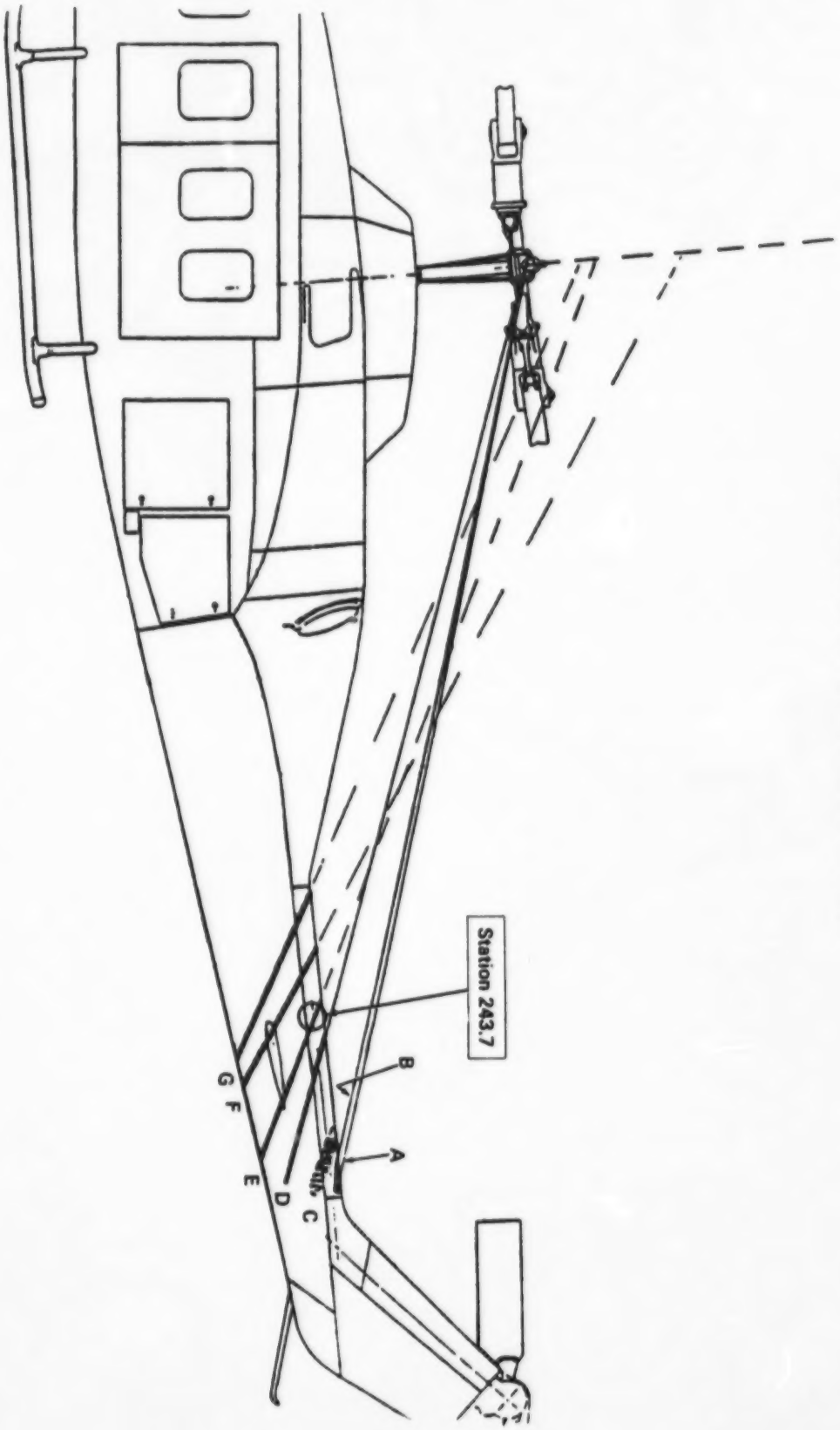
sequence either directly or indirectly. The maintenance- or inspection-related anomalies detailed in the majority report may have contributed to an initiating mechanical failure in some undetermined way.

In summary, we believe that the cause of this accident has not been determined, and we can not rule out some connection with the unexplained Bell 214ST accident which occurred on 13 March 1985 near Argentia, Newfoundland. Nevertheless, we concur with the recommendations of the majority.

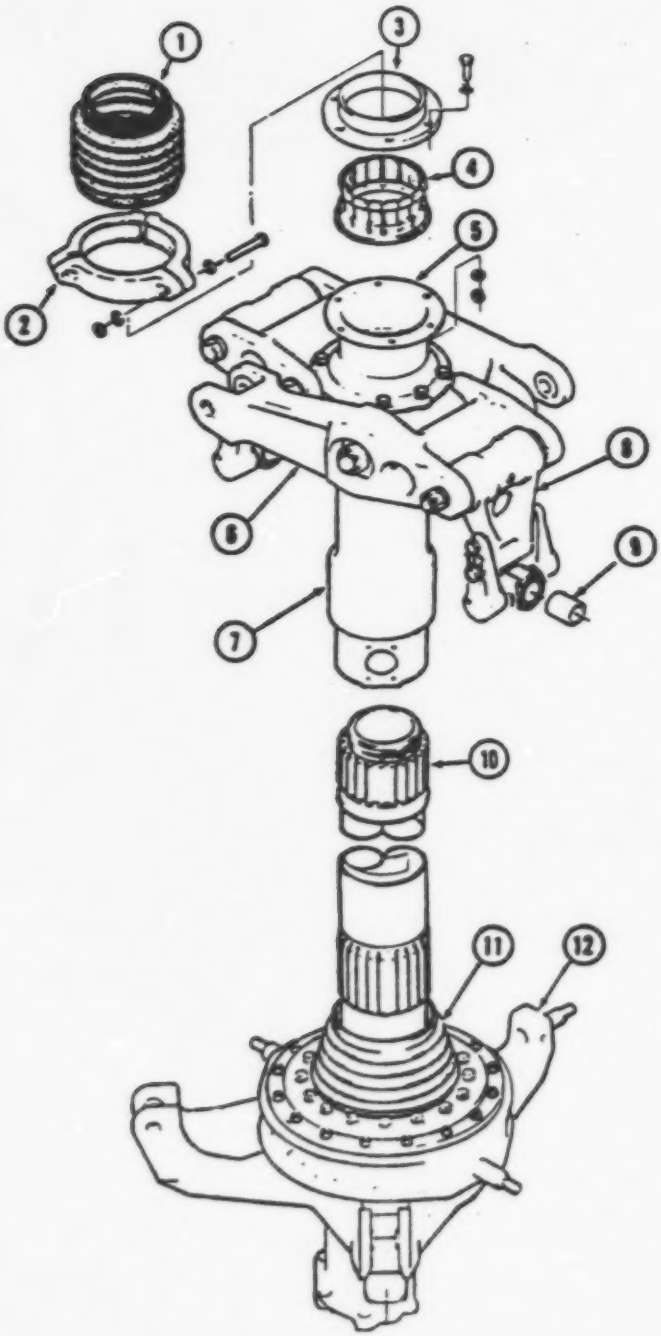
NOTE:

While some Members do not believe that the causes of the accident have been determined, the Board is unanimous in its identification of safety deficiencies and in its support of the proposed safety actions.

BLADE STRIKE LOCATION



MAIN ROTOR MAST ASSEMBLY



1. Boot
2. Clamp set
3. Ring
4. Friction collet set
5. Splined drive plate
6. Scissors assembly
7. Sleeve assembly
8. Drive link
9. Inner race
10. Mast
11. Boot
12. Non-rotating inner ring

APPENDIX C

LABORATORY REPORTS

The following laboratory reports were completed:

LP 256/85	COLLECTIVE SLEEVE FAILURE
LP 257/85	FIELD INVESTIGATION
LP 267/85	COLLECTIVE FRICTION COLLET
LP 283/85	EXAMINATION OF FLIGHT CONTROL SERVOS
LP 284/85	LONGERON FITTINGS
LP 285/85	LIGHT BULBS
LP 292/85	INPUT SHAFTS AND FLANGES
LP 33/86	MAST WEAR ANALYSIS
LP 59/86	TAIL ROTOR COUPLING FRACTURE
LP 67/86	AIRCRAFT INSTRUMENT AND SWITCH POSITION ANALYSIS
LP 75/86	FRACTURE MECHANICS CALCULATIONS
LP 151/86	IN-FLIGHT BREAKUP ANALYSIS
LP 197/86	PITCH LINKS ANALYSIS
LP 204/86	TRANSMISSION MOUNT AND SPRINGS
LP 38/87	MAIN ROTOR MAST

These reports are available upon request from the Canadian Aviation Safety Board.

GLOSSARY

AARS	attitude altitude retention system
AD	Airworthiness Directive
ADI	attitude display indicator
AFCS	automatic flight control system
agl	above ground level
ASB	Alert Service Bulletin
asl	above sea level
CASB	Canadian Aviation Safety Board
CMT	control motion transducer
CVR	cockpit voice recorder
FAA	Federal Aviation Administration
FAR	Federal Aviation Regulation
FBW	fly-by-wire system
FDR	flight data recorder
FMPI	fluorescent magnetic particle inspection
G	load factor
HEC	hydrogen embrittlement cracking
hr	hour(s)
HSI	horizontal situation indicator
IAS	indicated airspeed
IFR	instrument flight rules
IMC	instrument meteorological conditions
lat	latitude
lb	pound(s)
long	longitude
MAUW	maximum all-up weight
MDT	mountain daylight time
N	north
RFM	rotorcraft flight manual
SCAS	stability control augmentation system
SOR	Statement of Operational Requirements
UTC	Coordinated Universal Time
VFR	visual flight rules
Vne_{oei}	one-engine-inoperative never-exceed speed
VSI	vertical speed indicator
W	west
°	degrees(s)
'	minute(s)
"	second(s)